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NACA

RESEARCH MEMORANDUM

LONGITUDINAL STABILITY AND CONTROL CHARACTERISTICS AT TRANSONIC SPEEDS OF A SEMISPAN AIRPLANE MODEL HAVING A 45° SWEPTBACK WING AND TAIL AS OBTAINED BY THE TRANSONIC-BUMP METHOD

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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RESEARCH MEMORANDUM

LONGITUDINAL STABILITY AND CONTROL CHARACTERISTICS AT TRANSONIC SPEEDS OF A SEMISPAN AIRPLANE MODEL HAVING A 45° SWEPTBACK WING AND TAIL AS OBTAINED BY THE TRANSONIC-BUMP METHOD

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SUMMARY

An investigation has been made in the Langley high-speed 7- by 10-foot tunnel using the transonic-bump method to determine the longitudinal stability and control characteristics at transonic speeds of a semispan airplane model having a 45° sweptback wing and tail.

The results of the investigation indicated an increase in the rate of change of pitching-moment coefficient with lift coefficient at a constant Mach number $\left(-\frac{\partial C_m}{\partial C_L}\right)_M$ through the transonic range that was attributed to a rearward shift of the wing-fuselage aerodynamic-center location at subsonic speeds and to a rapid decrease in downwash at supersonic speeds.

At a Mach number of about 0.95 a moderate decrease occurred in both the lift-curve slope and in the stabilizer effectiveness. The high angle of sweep was effective in delaying the drag rise at zero angle of attack up to a Mach number of about 0.95.

The curve of stabilizer incidence required for trim against Mach number had an unstable variation between a Mach number of 0.90 and 1.20, but trim could be maintained throughout the Mach number range with a stabilizer deflection of only slightly more than 1°.

INTRODUCTION

Tests were made by the transonic-bump method to determine the longitudinal stability and control characteristics in the transonic range of a semispan airplane model having a 45° sweptback wing and tail. The tail was placed directly behind the wing for these tests. The tests were made through a Mach number [redacted] 0.50 to 1.23.

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SYMBOLS

C_L lift coefficient (Lift/qS)

C_D drag coefficient (Drag/qS)

C_m pitching-moment coefficient (Pitching moment/$qS\bar{c}$)

q free-stream dynamic pressure, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

S wing area, square feet

$\bar{c}$ wing mean aerodynamic chord, M.A.C., feet

ρ air density, slugs per cubic foot

V airspeed, feet per second

M test Mach number

M_l local air-stream Mach number

α angle of attack, degrees

R Reynolds number

i_t stabilizer incidence, degrees

ϵ downwash angle, degrees

$\frac{q_t}{q}$ ratio of effective dynamic pressure at tail to free-stream dynamic pressure

W airplane weight, pounds

h altitude, feet

a.c. aerodynamic center location, percent M.A.C.

C_{L_α} rate of change of lift coefficient with angle of attack

$\frac{\partial C_m}{\partial C_L}$ rate of change of pitching-moment coefficient with lift coefficient at constant Mach number

$\frac{\partial \epsilon}{\partial \alpha}$ rate of change of downwash angle with angle of attack

$\frac{\partial C_m}{\partial i_t}$ rate of change of pitching-moment coefficient with stabilizer incidence

MODEL AND APPARATUS

A three-view drawing of the semispan airplane model is given in figure 1 and the geometric characteristics are given in table I.

The tests were made in the Langley high-speed 7- by 10-foot tunnel by the transonic-bump method which involves placing a small semispan airplane model in the high-velocity-flow field generated over a curved surface. This method of testing is fully described in reference 1. A photograph of the model and the transonic-bump installation is shown in figure 2.

The model was mounted on a strain-gage balance and the lift, drag, and pitching moment were measured with a calibrated galvanometer. The angle of attack was changed with a small electric motor and the angle was determined with a calibrated slide-wire potentiometer.

TESTS

The variation of Reynolds number with Mach number for these tests is shown in figure 3.

The Mach number distribution over the bump is shown in figure 4 and indicates that the chordwise variation of Mach number becomes erratic at the higher Mach numbers. The effect of this variation is indeterminate and might result in the masking or exaggeration of trim or stability changes.

No tares were applied to the data to account for the presence of an end plate on the model and jet-boundary corrections were neglected since the model was small with respect to the tunnel.

Tests were made through the Mach number range from 0.50 to 1.23 at various angles of attack for two stabilizer settings and with the tail off. The angle of attack ranged from -1° to 5°. The stabilizer settings were -3.36° and 2.92°.

The pitching-moment coefficients are referred to the quarter chord of the mean aerodynamic chord.

RESULTS AND DISCUSSION

The variation of lift, drag, and pitching-moment coefficient with Mach number for various angles of attack and tail settings is given in figures 5 to 7.

Lift curves for various Mach numbers as obtained from figure 5 are presented in figure 8. The variation of lift-curve slope C_{L_α} with Mach number (fig. 9) indicated an increased slope up to $M \approx 0.95$ and then a moderate decrease in slope. A theoretical determination of the effect of compressibility on C_{L_α} in the subsonic range for finite aspect ratios was made using the experimental value of 0.052 at $M = 0.6$. Close agreement with experiment was indicated in the subsonic range.

The drag rise for the tail-off condition at $\alpha = 0$ is delayed up to a Mach number of about 0.95 by the high angle of sweepback (fig. 6). This delay in drag rise is similar to that observed in other tests of models having the same angle of sweepback. The high drag in the subsonic range is probably caused by the existence of the end plate on the fuselage.

The variation of pitching-moment coefficient with lift coefficient was obtained for various Mach numbers (fig. 10) by cross-plotting from the basic data of figures 5 and 7. From these curves it is possible to determine the rate of change of pitching-moment coefficient with lift coefficient $\partial C_m/\partial C_L$, the downwash variation $\partial\epsilon/\partial\alpha$, and the stabilizer effectiveness $\partial C_m/\partial i_t$ at the various Mach numbers. These curves are presented in figure 11. There is an increase in $-\partial C_m/\partial C_L$ beginning at $M \approx 0.80$ that is attributable to a rearward shift in the wing-fuselage aerodynamic center up to a Mach number of about 0.95. Above this Mach number the wing-fuselage aerodynamic center becomes constant and the continued increase in $-\partial C_m/\partial C_L$ is a result of a rapid decrease in $\partial\epsilon/\partial\alpha$. The downwash at supersonic speeds is greatly reduced from its subsonic value.

A decrease in the stabilizer effectiveness $\partial C_m/\partial i_t$ beginning at $M \approx 0.90$ is evident. This is probably a result of a decrease in the tail-lift-curve slope (the tail, being similar to the wing, is assumed to have the same C_{L_α} variation) and possibly a reduction in the dynamic-pressure ratio q_t/q. It is also possible that the reduced $\partial C_m/\partial i_t$ may be aggravated by the fact that the Reynolds number of the tail is less than that of the wing and the Mach number in the region of the tail may be slightly less than that of the wing.

Using the data of figure 10 and assuming a linear variation of pitching moment with stabilizer deflection, the variation of the stabilizer incidence required for trim against Mach number was determined for a hypothetical airplane similar to the model having a wing loading of 50 pounds per square foot and flying at an altitude of 30,000 feet. The airplane lift coefficient for this wing loading and altitude (fig. 12) was used in conjunction with figure 10 to obtain the stabilizer incidence required for trim through the Mach number range and lift-coefficient range shown in figure 13. A stable variation of stabilizer incidence required for trim with Mach number exists up to $M \approx 0.90$ but above that Mach number instability is indicated; that is, an increase in Mach number or a decrease in lift coefficient must be accompanied by a negative control movement (downward movement of stabilizer leading edge) up to $M = 1.2$. Trim can easily be maintained through the Mach number range up to $M = 1.2$, however, with slightly more than 1° of stabilizer deflection.

CONCLUDING REMARKS

The results of tests made by the transonic-bump method of a semispan airplane model having a 45° sweptback wing and tail indicated an increase in the rate of change of pitching-moment coefficient with lift coefficient at a constant Mach number $\left(-\frac{\partial C_m}{\partial C_L}\right)_M$ through the transonic range that was attributed to a rearward shift of the wing-fuselage aerodynamic center at subsonic speeds and to a rapid decrease in downwash at supersonic speeds.

The drag rise at zero angle of attack with tail off was delayed to a Mach number of about 0.95 by the high angle of sweep. A moderate decrease in the lift-curve slope occurred at a Mach number of about 0.95 and the stabilizer effectiveness was reduced.

The curve of stabilizer incidence for trim against Mach number had an unstable variation between a Mach number of 0.90 and 1.20; however, trim could be maintained with slightly more than 1° of stabilizer deflection.

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REFERENCE

1. Schneiter, Leslie E., and Ziff, Howard L.: Preliminary Investigation of Spoiler Lateral Control on a 42° Sweptback Wing at Transonic Speeds. NACA RM No. L7F19, 1947.

TABLE I

GEOMETRIC CHARACTERISTICS OF TRANSONIC STABILITY MODEL

Wing:	
Area (semispan), sq in.	12
Semispan, in.	4.24
Mean aerodynamic chord, in.	2.83
Thickness of biconvex section, percent c	0.10
Incidence, deg	0
Chord, root, in.	2.83
Chord, tip, in.	2.83
Sweep, deg	45
Taper ratio	1.0
Aspect ratio	3.0
Dihedral, deg	0
Tail:	
Area (semispan), sq in.	3
Semispan, in.	2.12
Mean aerodynamic chord, in.	1.415
Thickness of biconvex section, percent c	0.10
Chord, root, in.	1.415
Chord, tip, in.	1.415
Sweep, deg	45
Taper ratio	1.0
Aspect ratio	3.0
Dihedral, deg	0
Fuselage:	
Length, in.	9.75
Maximum diameter, in.	1.00

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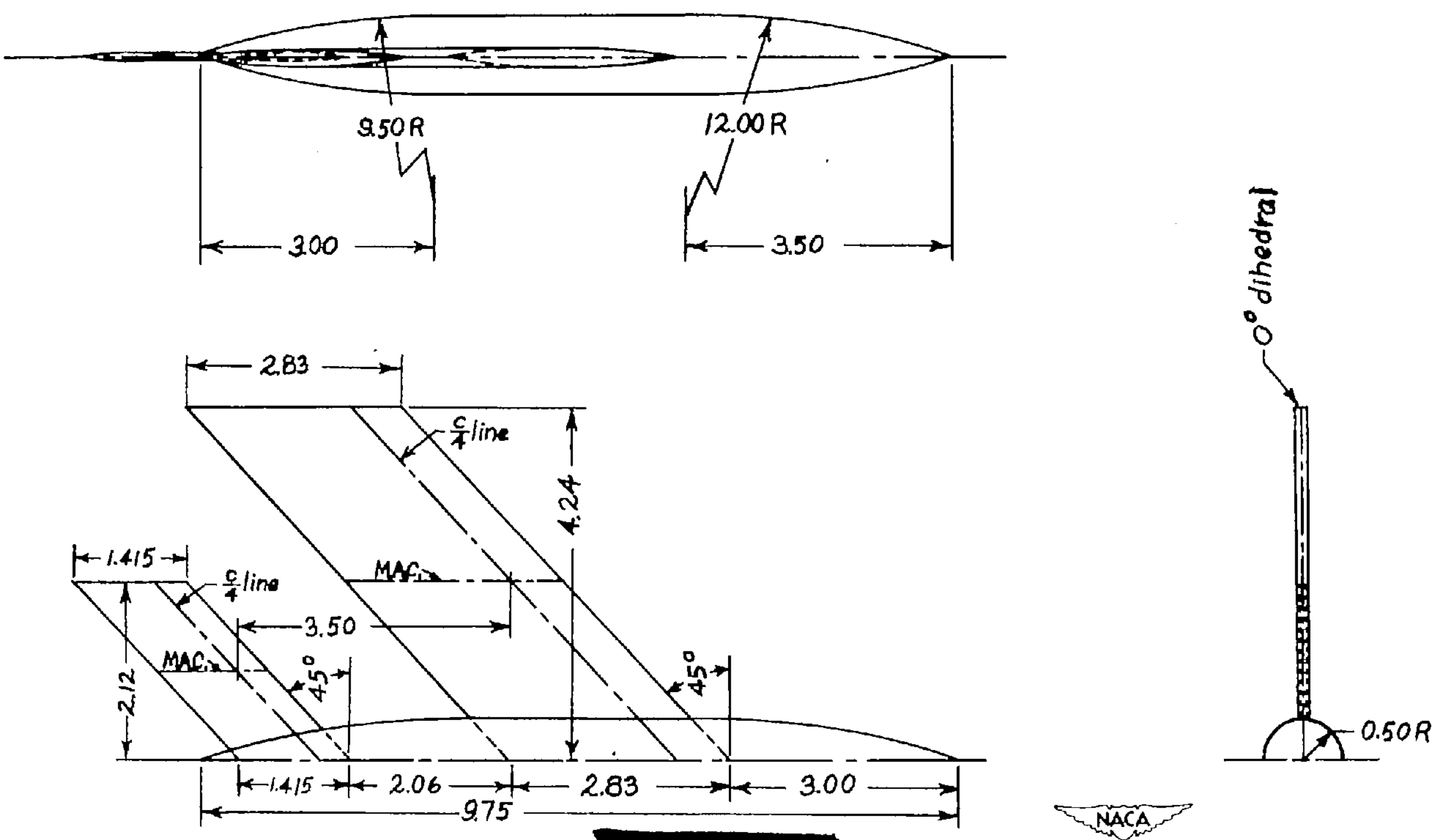

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Figure 1.- Three-view drawing of transonic stability model. Dimensions in inches except where noted.

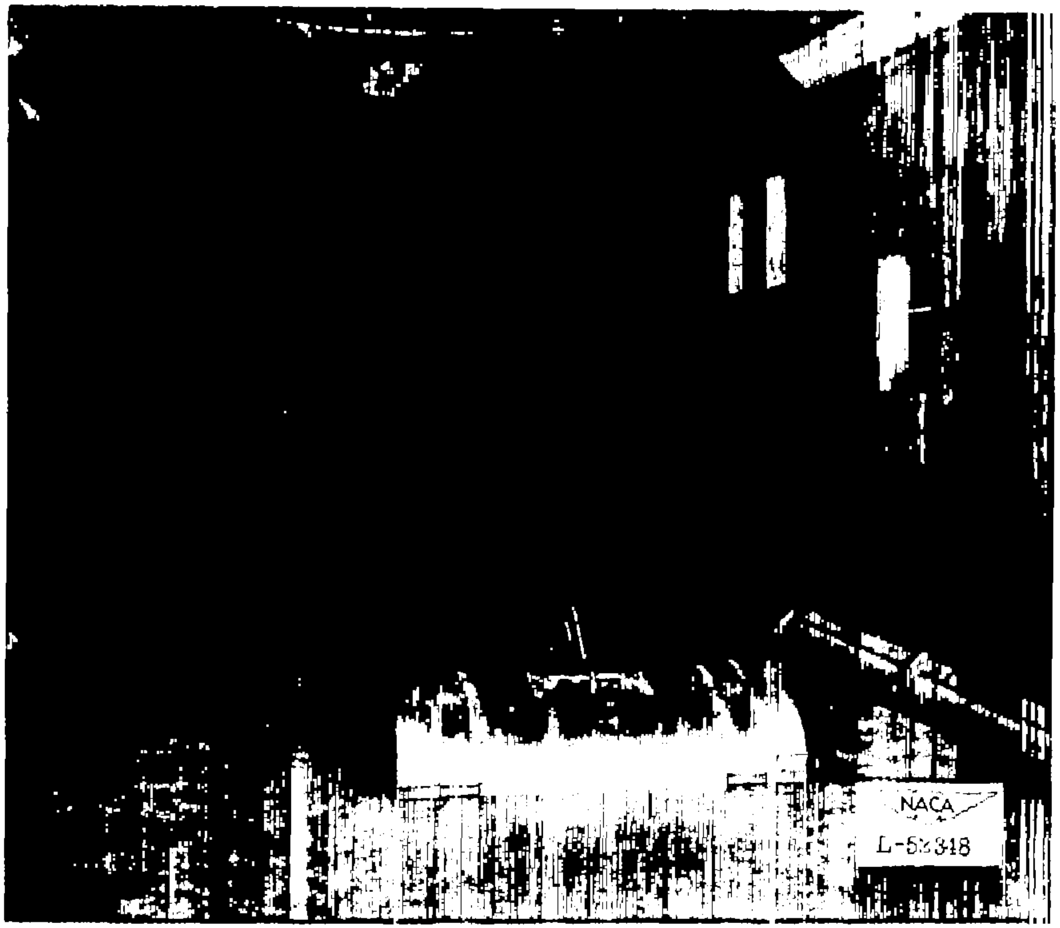

Figure 2.- Transonic bump and model installation in the Langley high-speed 7- by 10-foot tunnel.

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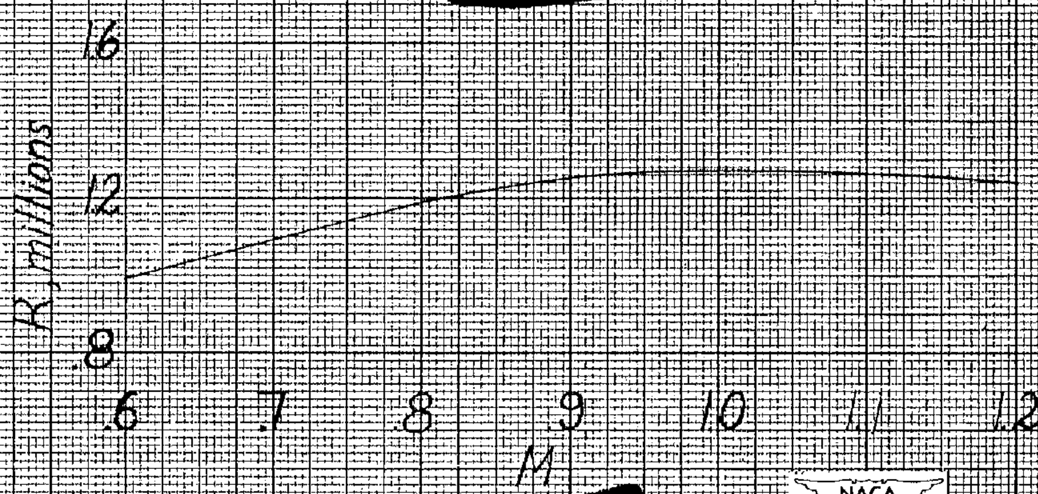

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Figure 3.- Variation of Reynolds number with test Mach number.

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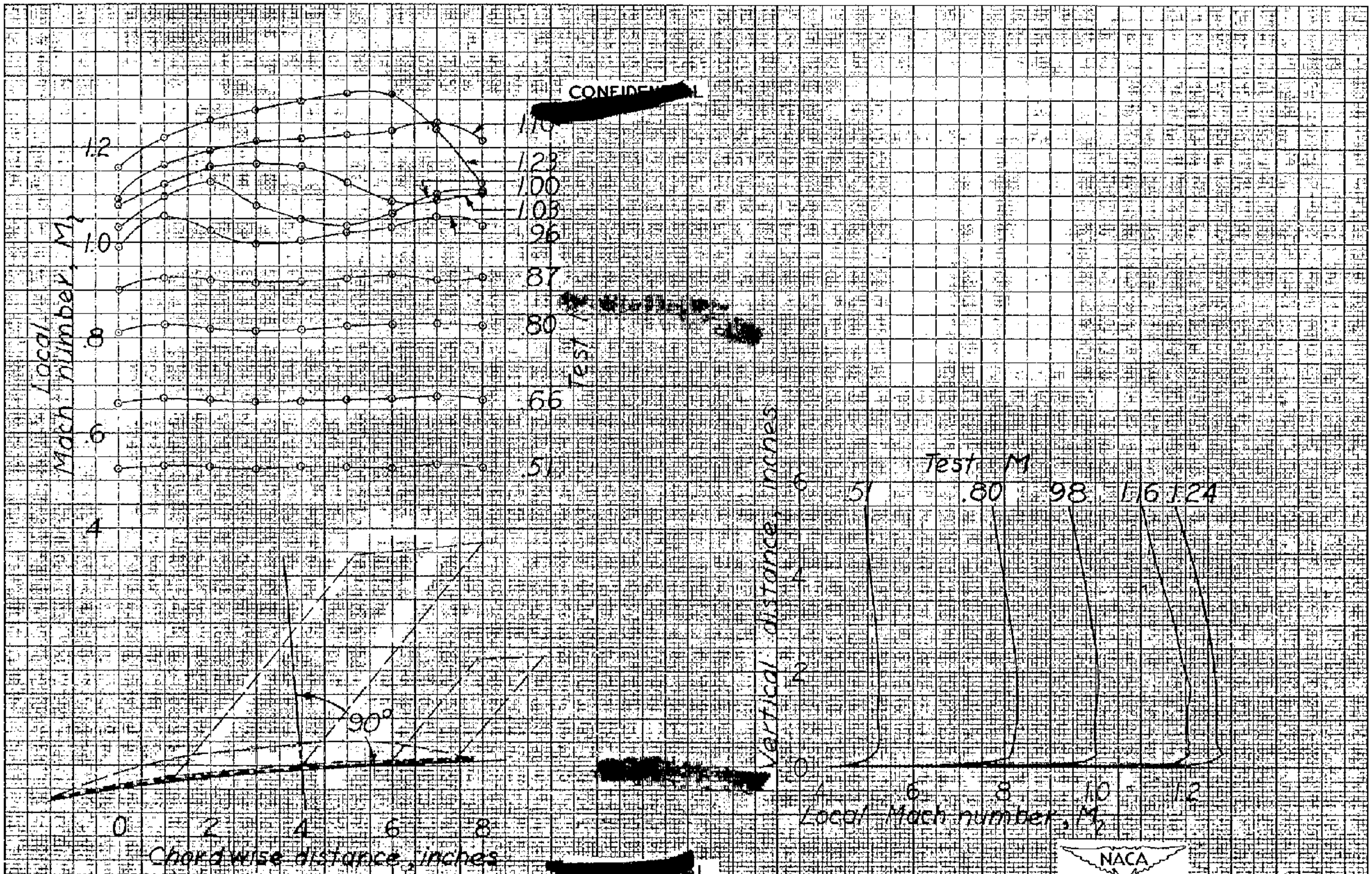

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Figure 4.- Spanwise and chordwise distribution of Mach number over transonic bump. Model off.

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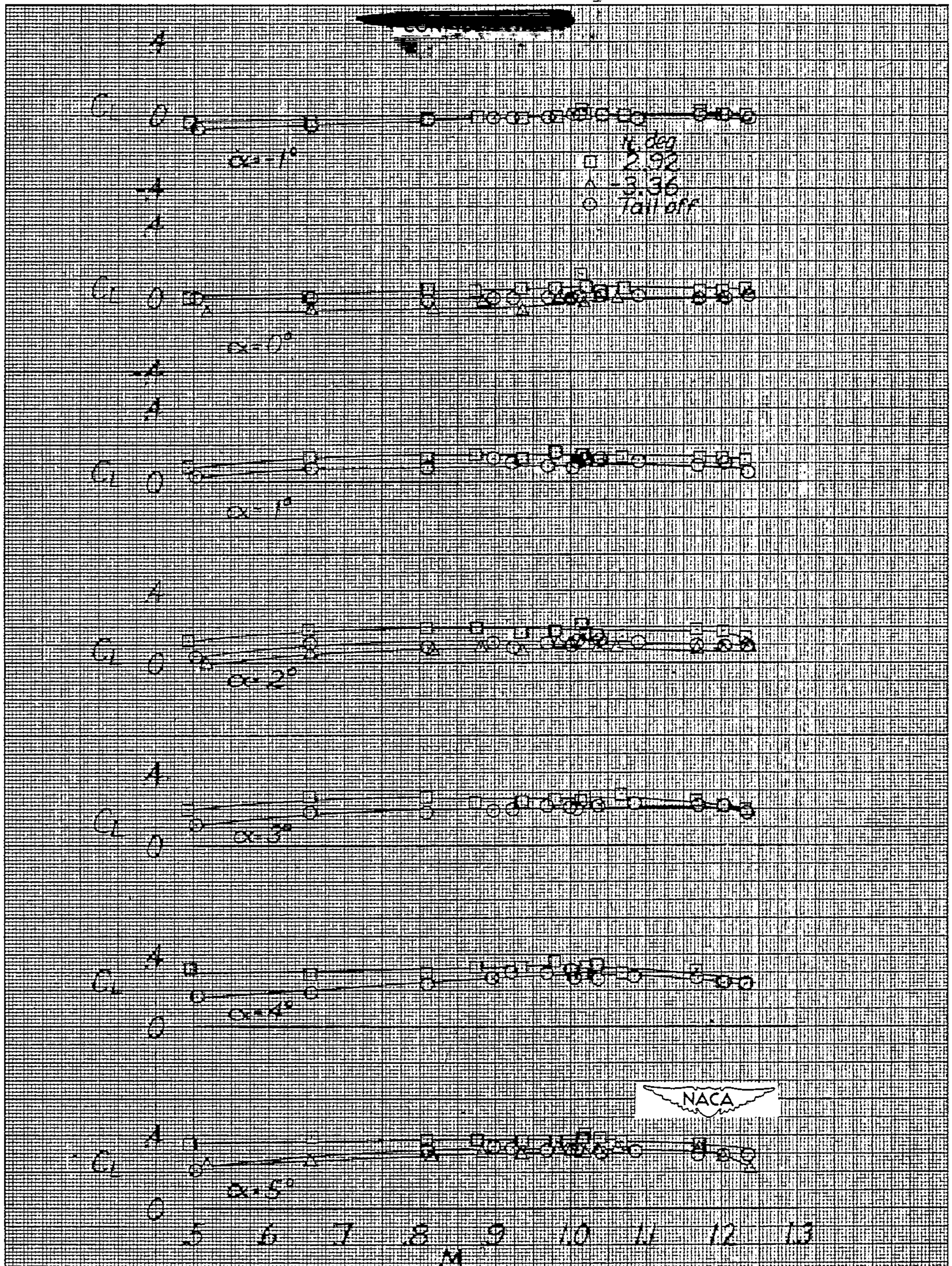

Figure 5.- Variation of lift coefficient with Mach number and angle of attack.

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[redacted]

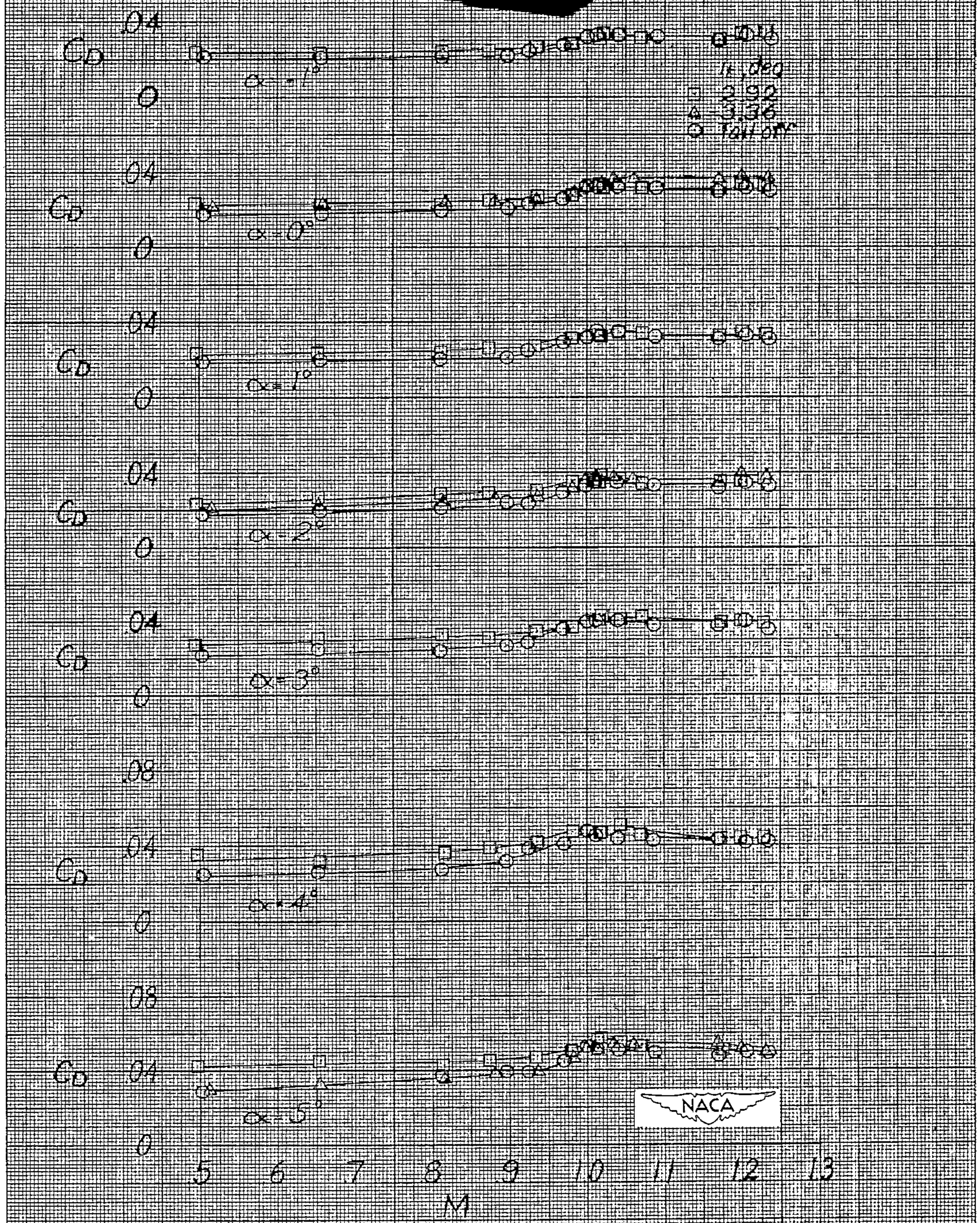

Figure 6. Variation of drag coefficient with Mach number and angle of attack.

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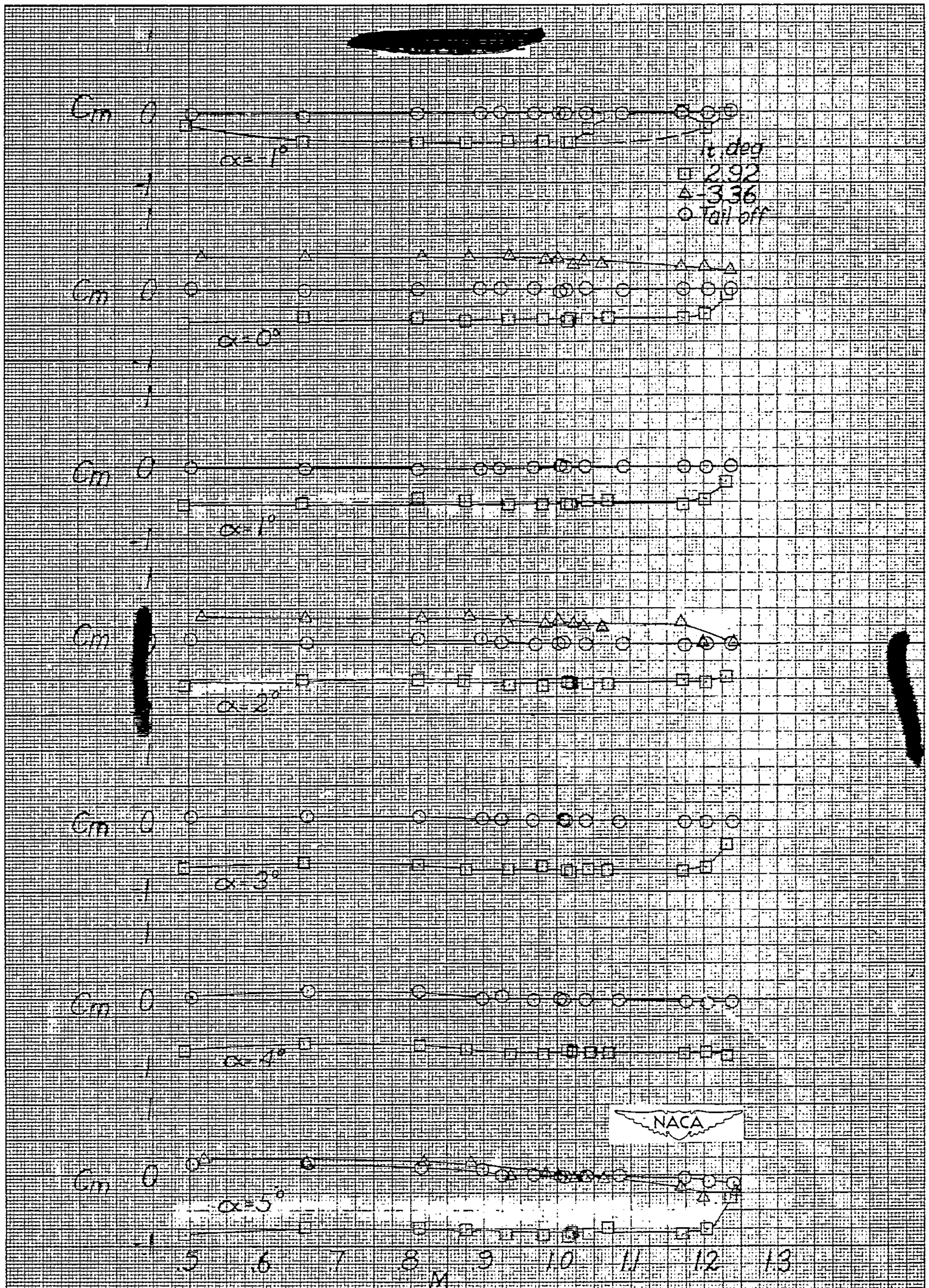

Figure 7.- Variation of pitching moment coefficient with Mach number and angle of attack.

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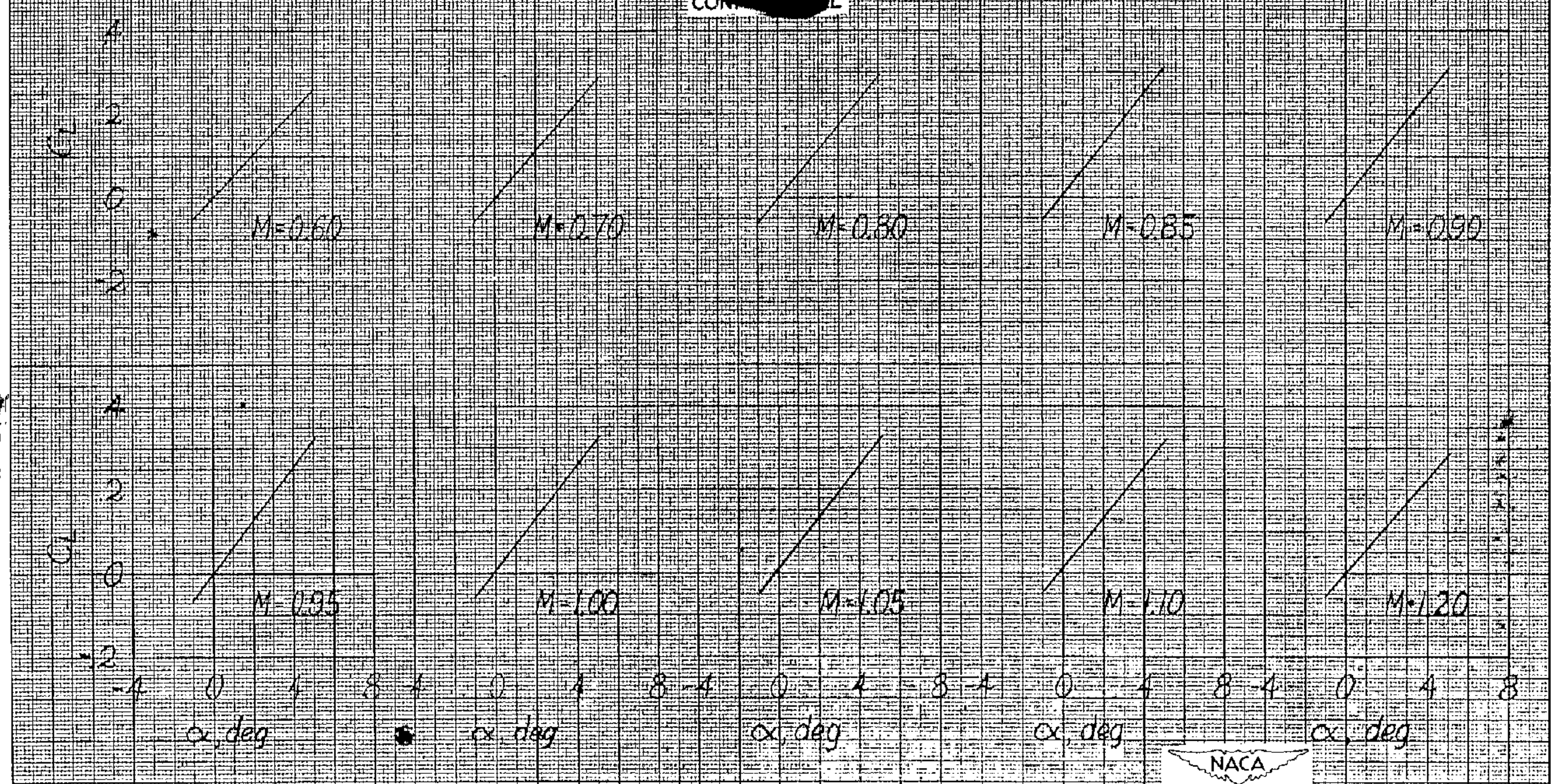

Figure 8.- Variation of lift coefficient with angle of attack for various Mach numbers. Tail off.

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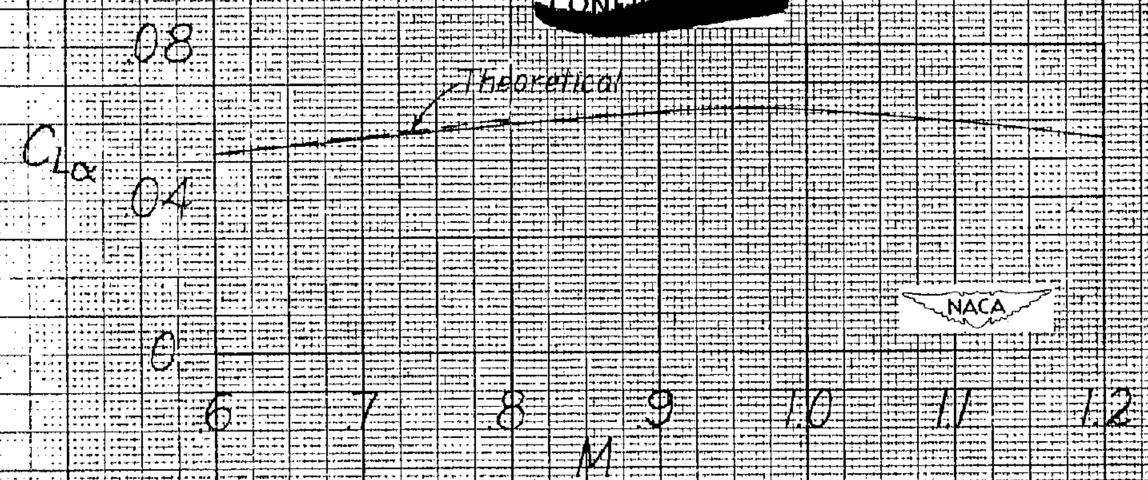

Figure 9.- Variation of lift-curve slope with Mach number.

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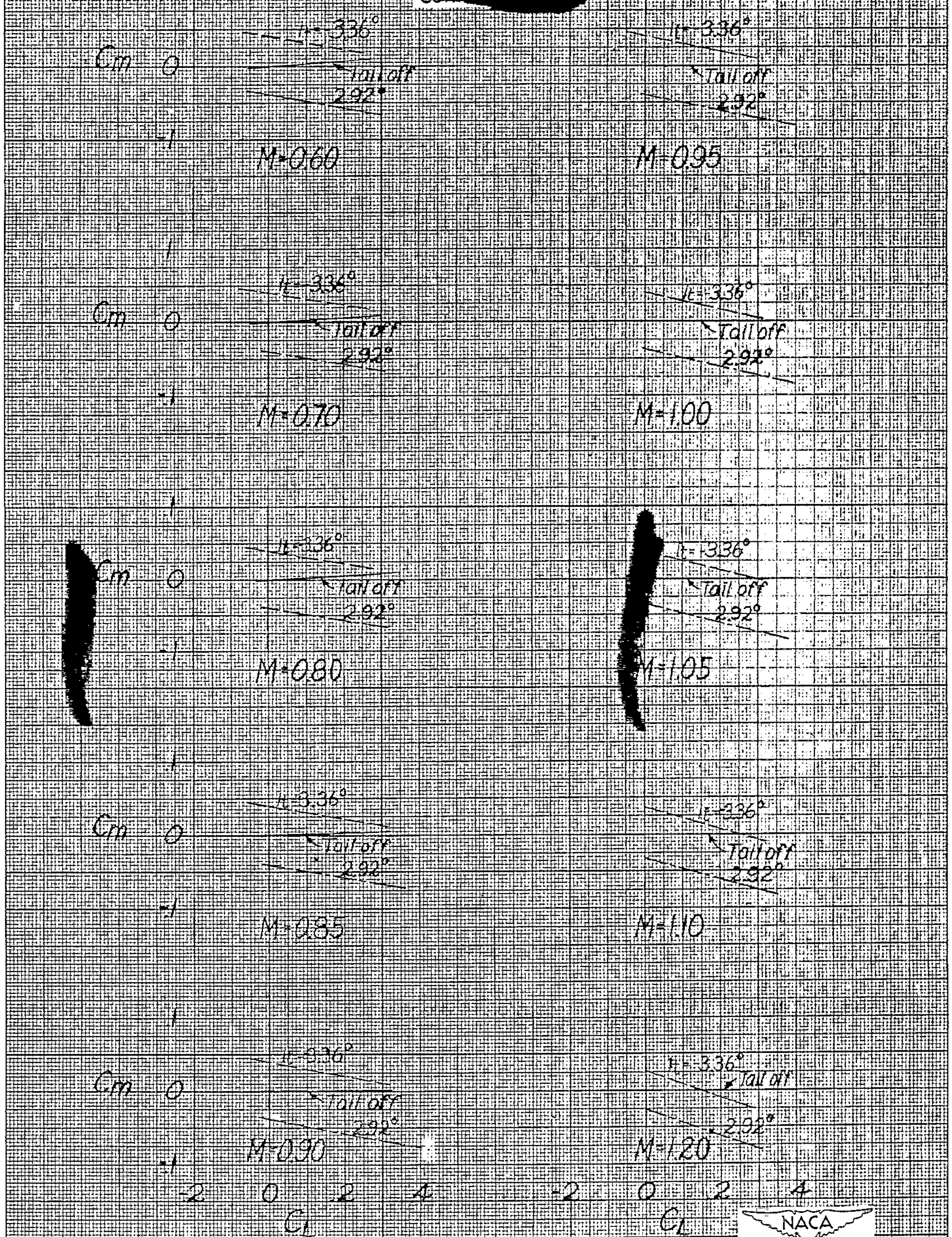

Figure 10.- Variation of pitching moment coefficient with lift coefficient for various Mach numbers and stabilizer deflections.

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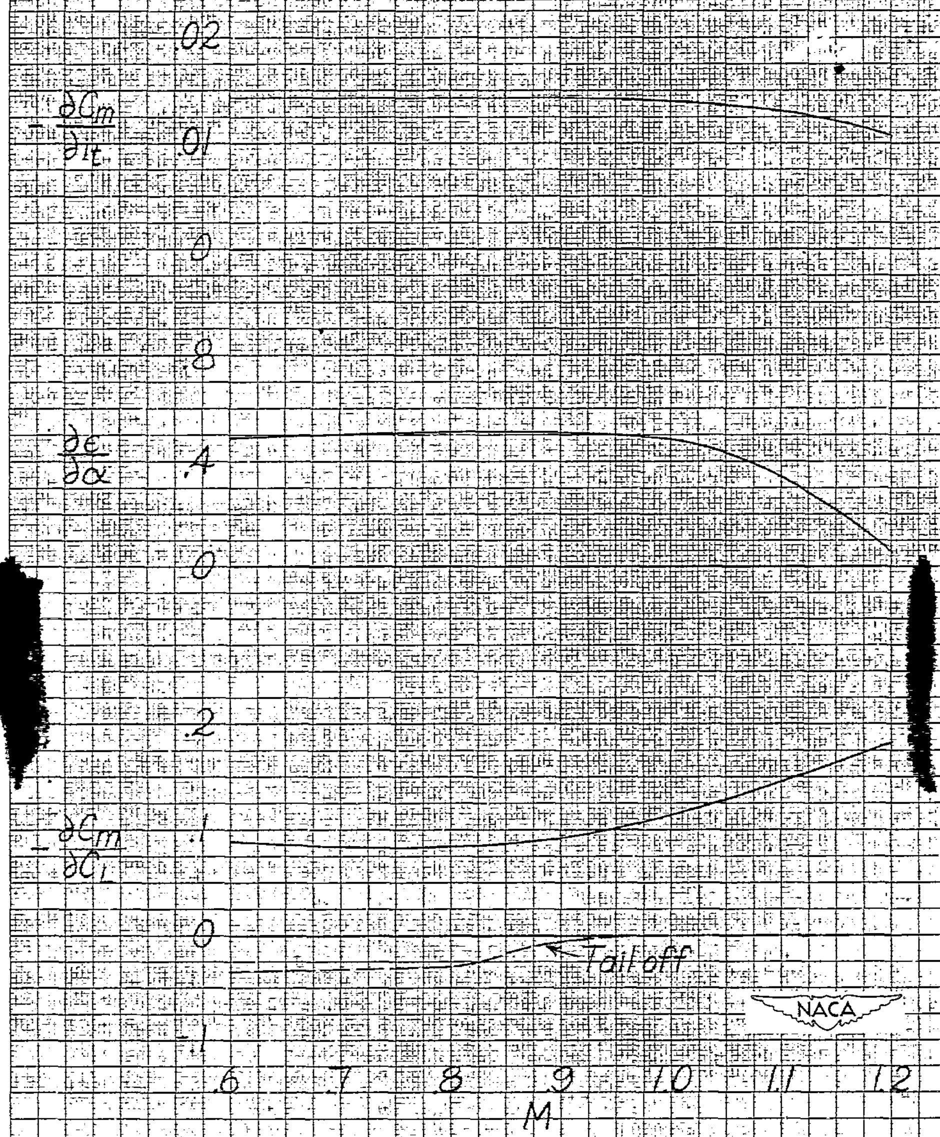

Figure 11.- Variation of stabilizer effectiveness, downwash, and static longitudinal stability with Mach number.

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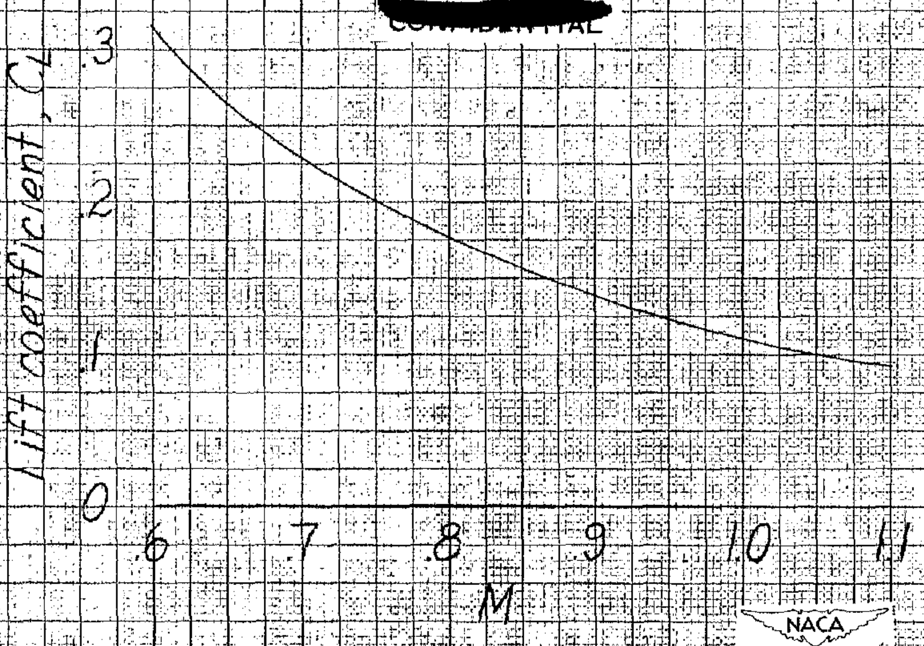

Figure 12.- Variation with Mach number of the lift coefficient for level flight. W/S = 50 lb/sq ft; h = 30,000 ft.

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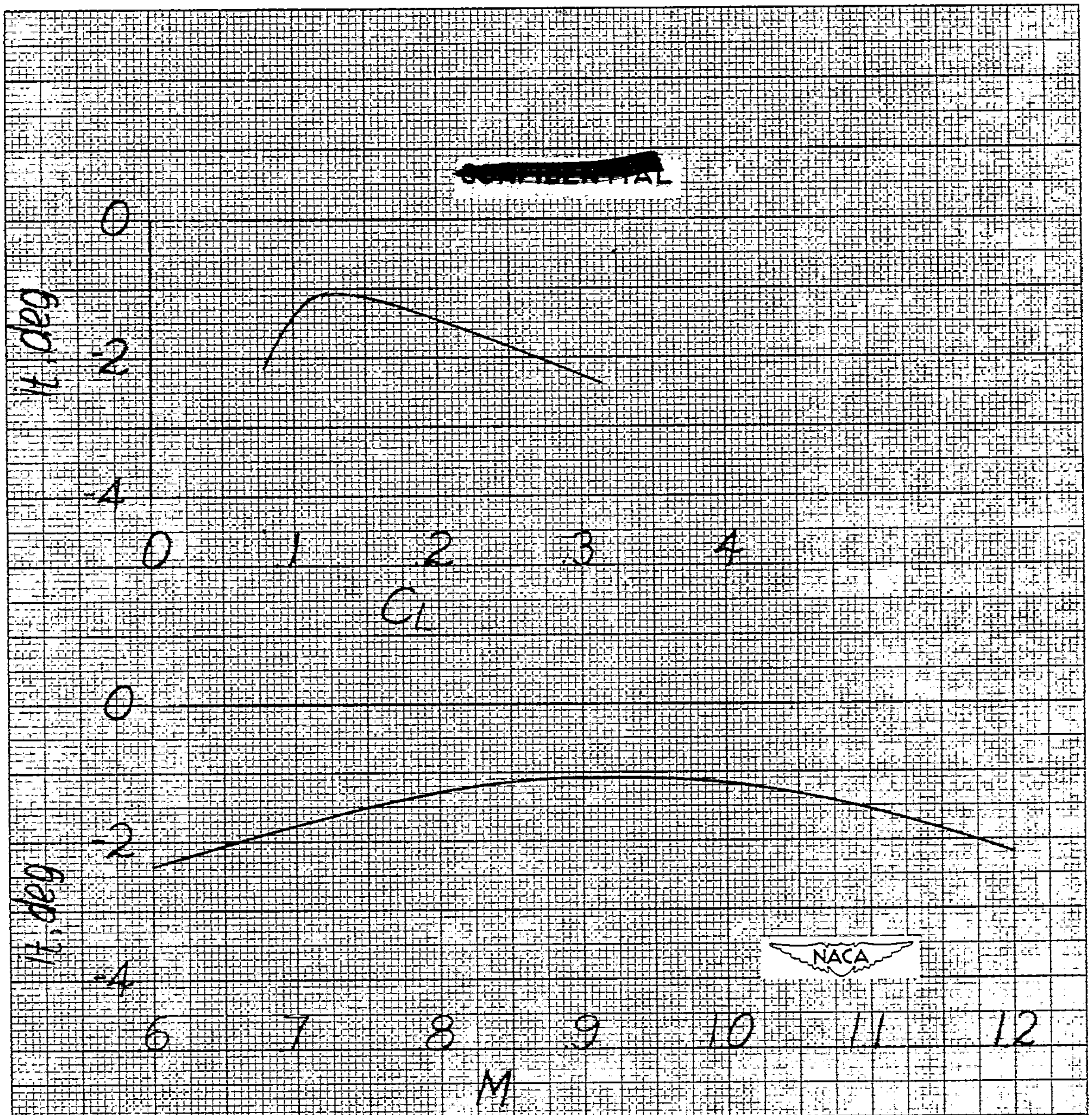

Figure 13.- Variation of stabilizer incidence required for trim with lift coefficient and Mach number. W/S = 50 lb/sq ft, h = 30,000 ft. [redacted]